CHARACTERIZATION OF FATIGUE BEHAVIOR OF AIRCRAFT FUSELAGE STRUCTURES

John G. Bakuckas, Jr.*, Catherine A. Bigelow*, and Paul W. Tan*

This paper presents representative results from an ongoing experimental and analytical study of the fatigue behavior of aircraft fuselage structure. Several areas of aircraft structural integrity research have been undertaken, including the initiation and development of multiple-site damage (MSD), effects of MSD on the residual strength behavior, and methods to reduce fatigue-related problems using polyisocyanurate foam. The Full-Scale Aircraft Structural Test Evaluation and Research facility was used to identify and track the damage evolution process and obtain key data for model calibration and validation. Computational simulations compared well with the experimental observations in terms of strain distributions and crack growth characteristics. Results show that the majority of fatigue life was spent in initiating cracks. Cracks initiated from the inner-faying surface at rivet holes in the outermost fastener row in the lap joints and progressed through the thickness. Once first linkup occurs, crack growth rate increased substantially. Although small multiple cracking did not have an effect on the overall global strain response, it significantly reduced the fatigue life and residual strength.

INTRODUCTION

A major focus of the structural integrity research supporting the Federal Aviation Administration's (FAA) National Aging Aircraft Research Program has been the assessment of fatigue mechanisms in aircraft structure through computational and experimental analysis. Emphasis has been placed on determining the causes, growth mechanisms, and consequences of widespread fatigue damage (WFD). Knowledge of multiple-site damage (MSD) nucleation time, pattern, and distribution, as well as its subsequent growth and effects on residual strength, is a prerequisite for planning an acceptable program to preclude the occurrence of WFD.

* FAA William J. Hughes Technical Center, AAR-400, Airport and Aircraft Safety Research and Development Division, Atlantic City International Airport, NJ 08405, USA.

As part of the FAA's core capability, a unique, state-of-the-art Full-Scale Aircraft Structural Test Evaluation and Research (FASTER) facility has been established at the FAA William J. Hughes Technical Center for testing large curved panels representative of aircraft fuselage structure. This facility provides experimental data to support and validate analytical methods under development, including WFD prediction, repair analysis and design, and new aircraft design methodologies. The fixture, shown in Figure 1, is designed to simulate the actual loads an aircraft fuselage structure is subjected to while in flight, including differential pressure, longitudinal load, hoop load in the skin and frames, and shear load. Both quasi-static and long-term durability spectrum loadings can be applied in the FASTER facility. A key component of the FASTER facility is the Remote Controlled Crack Monitoring (RCCM) system developed to track and record the formation and growth of multiple cracks in real time during a test. A full description of the FASTER facility is provided in references 1 and 2.

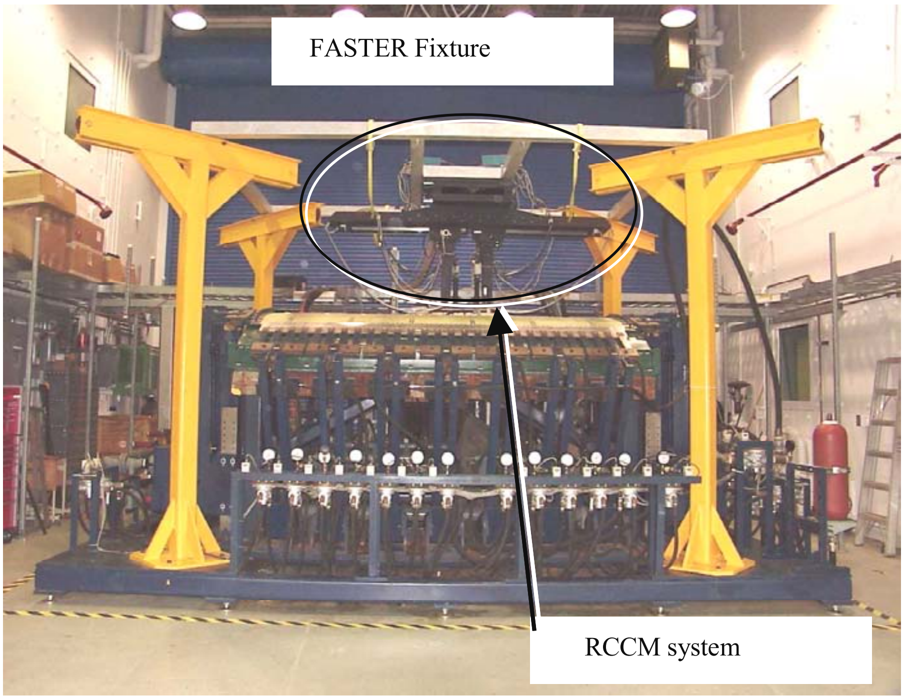


Figure 1. Full-Scale Aircraft Structural Test Evaluation and Research facility

Several programs have been undertaken to investigate the fatigue and residual strength characteristics of fuselage structure. A variety of fuselage panels has been tested and analyzed, including undamaged panels and panels manufactured with crack-like slits to simulate initial damage scenarios. In addition, a panel with polyisocyanurate foam was tested to assess its effect on fatigue behavior. For each panel tested, strain surveys were first conducted to ensure proper load transfer from the load application points to the panels. Fatigue damage was quantified in terms of crack initiation, crack distribution (location and size), and subsequent growth using the RCCM and other nondestructive inspection methods. Residual strength tests were conducted to determine the effects of damage states on load-carrying capacity.

Results presented include comparisons of strain distributions, fatigue crack growth characteristics, and the damage growth process for the panels tested. In general, the majority of fatigue life was spent in initiating and forming cracks. Although multiple cracking did not have an effect on the overall global strain response, it significantly reduced the fatigue life and residual strength. In addition, the application of polyisocyanurate foam to fuselage panels was effective, enhancing crack growth performance.

EXPERIMENTS

Table 1 lists the major test programs undertaken recently, using the FASTER facility with the test objective, the panel description, the initial damage state, and the test type.

Typical Panel Configuration

The panels tested represented narrow-body fuselage structure consisting of skin, frames, shear clips, stringers, and either longitudinal splice or circumferential joints. Typical panel dimensions are 120" in the longitudinal direction, 68" in the circumferential direction, with a radius of 66" and skin thickness of 0.063". Each panel has six frames with a 19" spacing and seven stringers with a 7.5" spacing. Along the perimeter, reinforcing doublers with a length of 112" on the longitudinal sides and 56" on the hoop sides were added. There were 28 load application points on each longitudinal side and 16 load application points on each hoop side. Panels were typically instrumented with 64 channels of strain gages for strain surveys.

Panels CVP1 through CVP4 and CVPB contained longitudinal lap joints consisting of two layers of the 2024-T3 panel skin and two layers of 2024-T3 finger doublers, Figure 2. Four rows of fasteners, A, B, C, and D were used to connect the skin and doublers. Panels CVP3 and CVP4 contained a circumferential butt joint between frames F3 and F4. Further details of the panel configurations are provided in reference 2.

Crack Initiation Test Panel

The purpose of this test program was to study fatigue crack formation, growth, and distribution in a fuselage panel [3]. Panel CVPB contained a longitudinal lap splice as shown in Figure 2. The panel was pristine with no initial damage. A strain survey was conducted to ensure proper load introduction and then subjected

Table 1. Test Matrix

Program Name and Objective	Panel Designation and Description	Test Type
Crack Initiation Test: Study fatigue crack initiation and MSD evolution and distribution	CVPB: Longitudinal splice. Pristine panel.	- Strain survey - Fatigue crack growth - Residual Strength
MSD Panel Tests: Determine the effects of multiple cracks on the fatigue crack growth and residual strength	CVP1: Longitudinal splice with lead crack in outer critical rivet row of joint. Severed central frame.	- Strain survey - Fatigue crack growth - Residual Strength
	CVP3: Circumferential butt joint with lead crack in outer critical rivet row of joint. Severed central stringer.	- Strain survey - Fatigue crack growth - Residual Strength
	CVP2: Longitudinal splice with lead crack and collinear MSD in critical rivet row of joint. Severed central frame.	- Strain survey - Fatigue crack growth - Residual Strength
	CVP4: Circumferential butt joint with lead crack and MSD in critical rivet row of joint. Severed central stringer.	- Strain survey - Fatigue crack growth - Residual Strength
Fatigue Enhancement Test: Assess polyisocyanurate polymer foam on fuselage response	CVPAP: Longitudinal lap splice removed from retired DC-9. As-Received, Mid-bay crack.	- Strain survey - Fatigue crack growth

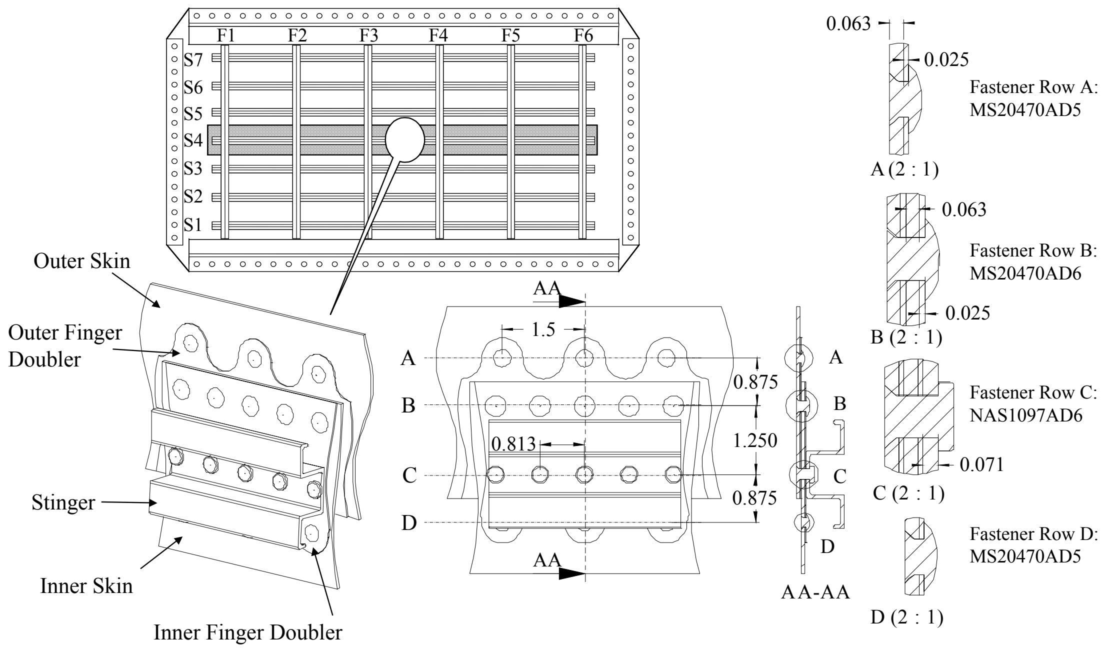


Figure 2. Lap joint construction along stringer S4. All dimensions in inches

to constant-amplitude fatigue loading. During fatigue loading, a rotating eddy-current probe was used to inspect for nonvisual cracks that develop under rivet heads in the lap joint area.

MSD Panel Test Panels

Panels CVP1 through CVP4 were tested to determine the effects of multiple cracks on the fatigue crack growth and residual strength of fuselage structure with a large lead crack [2, 4]. Initial damage scenarios were inserted into the longitudinal lap joint panels, as shown in Figure 3. Panel CVP1 contains a longitudinal lap splice with a lead crack. Panel CVP2 has the same configuration and lead crack as CVP1 with the addition of multiple, small cracks emanating from rivet holes ahead of the lead crack. For circumferential butt joint panels CVP3 and CVP4, similar initial flaw scenarios were machined in the outer critical rivet row of the butt joint. Details are provided in reference 2.

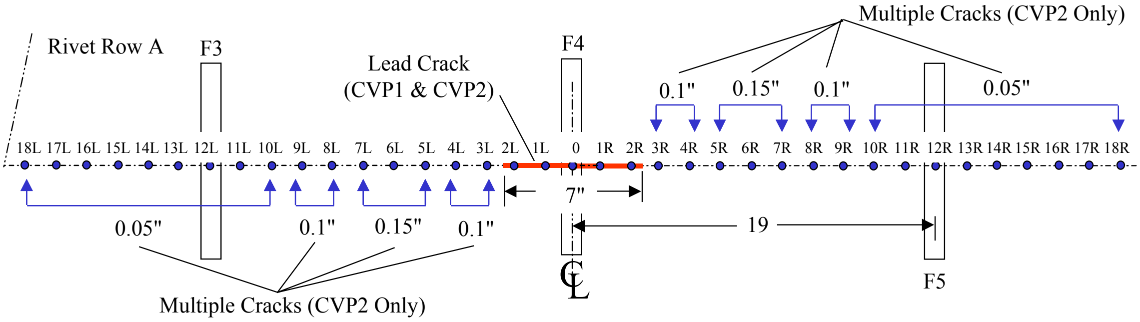


Figure 3. Initial damage scenarios for MSD Panel Test Program

Fatigue Enhancement Test Panel

The purpose of this test program was to assess the fatigue properties of fuselage structure with polyisocyanurate polymer foam. Panel CVPAP was extracted from a retired narrow-body aircraft and contained a longitudinal lap splice. The skin was 2014-T6 aluminum with a thickness of 0.05". Initially, the panel was tested in the as-received condition to obtain baseline data. A 5.5" mid-bay crack was inserted in the panel to measure crack bulging deflection and the fatigue crack growth. Afterwards, the mid-bay crack was repaired and a 3½-inch-thick layer of foam was applied to the inner surface of the panel, as shown in Figure 4. Another 5.5" mid-bay crack was inserted in the panel. Strain, crack-bulging deflection, and the fatigue crack growth were then measured and compared to the baseline data.

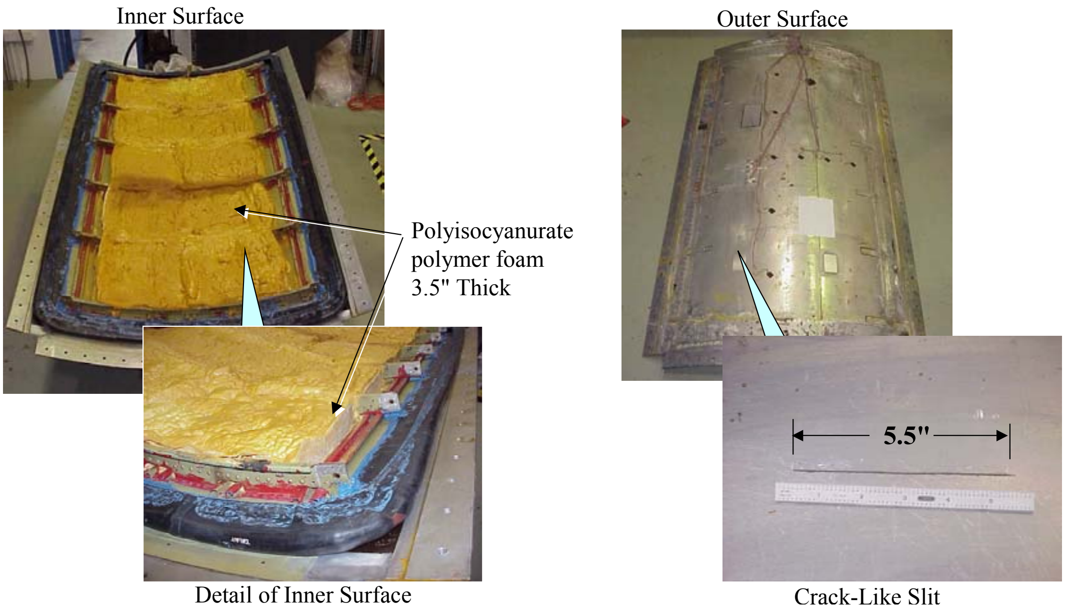


Figure 4. Panel CVPAP with polyisocyanurate polymer foam

Test Procedures

The panels were subjected to the applied loadings listed in Table 2 for strain survey, fatigue crack growth, and residual strength tests. For the longitudinal lap joint panels (CVPA, CVPAP, CVP1, CVP2, and CVPB), the applied load simulated the cylindrical pressurization that a section of the fuselage along the neutral axis would experience. For the circumferential butt joint panels (CVP3 and CVP4), the applied load simulated the fuselage down-bending condition that a fuselage section would experience along the crown of the aircraft where the longitudinal stress is 50% higher than the hoop stress.

Table 2. Applied load components

Panel	Maximum Load			
	Pressure (psi)	Hoop (lb/in)	Frame (lb/in)	Long. (lb/in)
CVPAP	5.0	283.8	46.2	165.0
CVP1	10.1	554.6	111.9	333.3
CVP2	10.1	554.6	111.9	333.3
CVP3	8.8	483.2	97.6	875.7
CVP4	8.8	483.2	97.6	875.7
CVPB	16.0	878.6	177.4	528.0

For strain survey tests, quasi-static loadings were applied in ten equal increments up to the maximum loads listed in Table 2. For fatigue crack growth tests, constant amplitude loading was applied at a frequency of 0.2 Hz with an R-ratio (minimum to maximum load) of 0.1 using the maximum loads are listed in Table 2. For CVPB, 75% underload cycles were used to mark the fracture surfaces to reconstruct the crack growth histories. For all tests, growth of the lead crack and small multiple cracks was continuously monitored and recorded using the RCCM system. For the residual strength tests, the load was applied quasi-statically up to catastrophic failure proportional to the values listed in Table 2.

ANALYSIS

For each test program, geometric nonlinear finite element analyses were conducted to predict the strain distributions and to compute the stress-intensity factor (SIF) solutions. The panels were modeled using shell elements with each node having six degrees of freedom. Figure 5 shows the global view of a typical finite element model of panels CVP1 and CVP2. Four-noded shell elements were used throughout to model the skin, frames, shear clip, stringers, and intercostals except near the crack tips. In the immediate vicinity of the crack tips, eight-noded shell elements

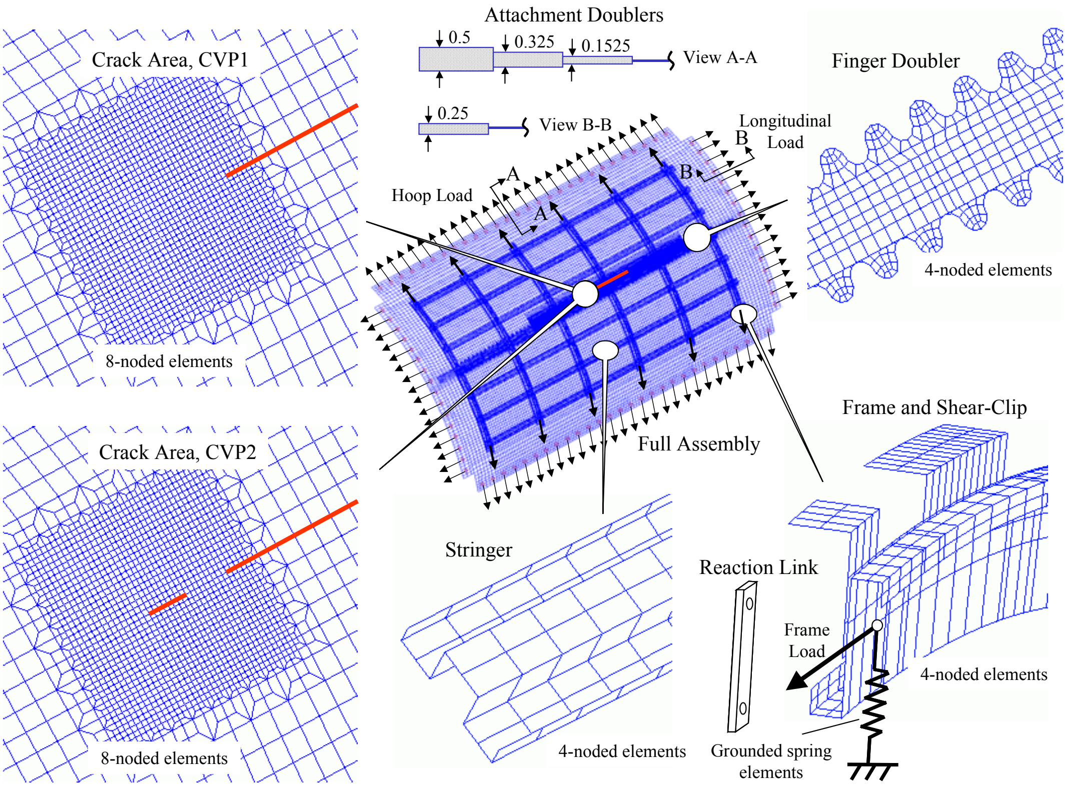


Figure 5. Finite element model of panel tested

were used. The model contained the major geometric details and dimensions of the panels, including the cross-section properties of the substructure (frames, stringers, shear-clip, intercostals), the finger doublers, and the load attachment doublers. Beam elements were used to model the rivets. Typically, the panel models had 250,000 degrees of freedom. The load conditions specified in Table 2 were simulated in the analysis. For the hoop, frame, and longitudinal loads, nodal point forces were applied at the load application points, as shown by the arrows in Figure 5. Internal pressure was applied to the inner surface of the skin.

Analysis of Crack Initiation and Small Cracks

The evolution of fatigue cracks at fastener holes in aircraft joint structure is complex due to unknown local stress states and the large number of uncertainties from physical processes including rivet clamp-up and fretting. Analysis from first principles where various factors affecting the initiation and growth of small cracks are decomposed and assessed is difficult and often impractical. In efforts to simplify analysis, an empirical engineering approach was developed as outlined in Figure 6 using the flat panel test data reported in reference 5. These flat panels had identical joint construction to the curved longitudinal lap joint panels listed in Table 1. It was assumed that the crack initiation and small crack growth behavior would be similar. In the approach, the SIF for the flat panels was estimated using the flat panel crack growth data and fatigue crack growth properties:

$$\Delta K_{flat} = \left(\frac{exp\left(A\,ln(a) - B \right)}{C} \right)^{-n} \qquad (1)$$

where the numerator of the ratio is the flat panel crack growth data represented in the following equation form:

$$\frac{da}{dN} = exp\left(A\,ln(a) - B \right) \qquad (2)$$

and C and n are the parameters of the Paris equation:

$$\frac{da}{dN} = C \Delta K^{n} \qquad (3)$$

The ratio of the local strain in the curved and flat panels at the same location near the joint was used as a transfer function β to relate the flat and curved panel SIF:

$$\Delta K_{curved} = \beta \Delta K_{flat} \qquad (4)$$

The crack growth of the curved panel was obtained using the curved panel SIF and the equivalent initial flaw size (EIFS) back calculated using the flat panel SIF.

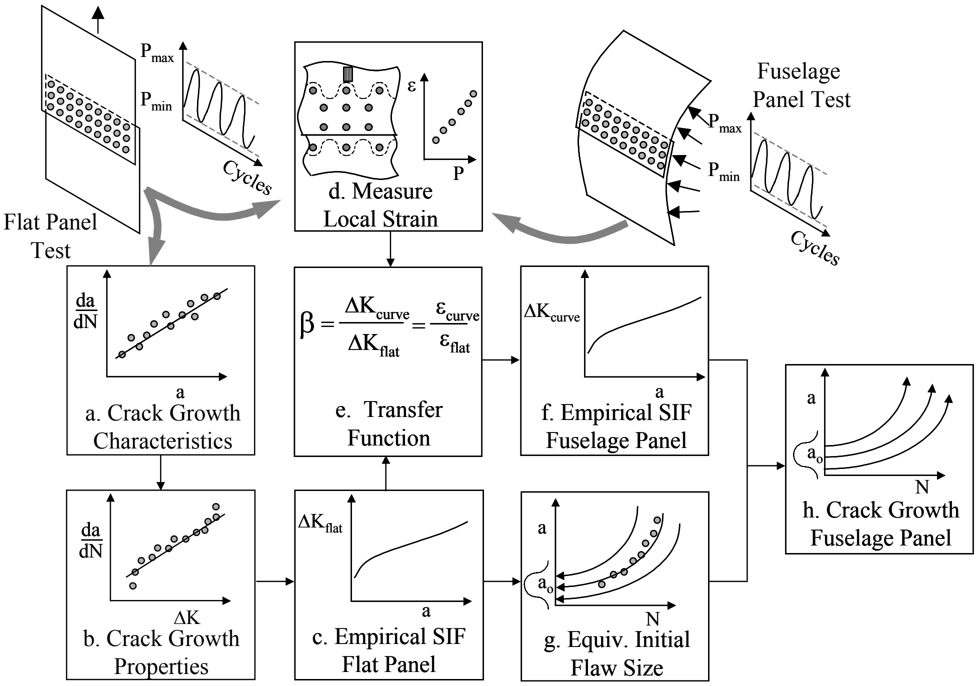


Figure 6. Crack initiation and small crack analysis approach

Analysis of Lead Cracks

The SIF solutions were used to predict the fatigue crack growth characteristics of the curved panels with a large lead crack. The effective SIF was calculated from finite element analyses and is defined as:

$$K_{eff} = \sqrt{E\left(\left(\frac{K_I^{\,2}}{E}\right)+\left(\frac{K_{II}^{\,2}}{E}\right)+\left(\frac{1+\nu}{3+\nu}\right)\left(\frac{\pi k_I^{\,2}}{3E}\right)+\left(\frac{1+\nu}{3+\nu}\right)\left(\frac{\pi k_{II}^{\,2}}{3E}\right)\right)} \qquad (5)$$

where K_I, K_{II}, k_I, and k_{II} are the mixed mode SIFs calculated, using the Modified Crack Closure Integral method [6]. The effective SIF was used along with the fatigue properties defined in Equation 3 to calculate the fatigue crack growth. Table 3 lists the parameters used in the analysis.

RESULTS AND DISCUSSION

The test programs listed in Table 1 included strain survey under quasi-static loading conditions to ensure proper load introduction, fatigue loading to measure crack formation and growth, and residual strength test under quasi-static load conditions to measure the load carrying capacity. Representative results from the test programs are outlined in the subsequent sections.

Table 3. Modeling parameter values

Parameter	Value
Crack Rate Slope, A	1.073568203
Crack Rate Intercept, B	10.07857622
Paris Intercept, C	4.5336E-10
Paris Slope, n	3.8
Modulus of Elasticity, E (ksi)	10500
Poisson's Ratio, ν	0.3
β, Load Transfer	1.14

Crack Initiation Test Results

The initiation, distribution, and linkup of MSD in the lap joint of an initially undamaged fuselage panel was studied. Test results during the strain survey revealed a high local-bending deformation along the critical outer rivet row in the lap joint area, the same area where MSD cracks initiated [3]. As shown in Figure 7, there is more local bending for the curved panel CVPB compared to the flat panel test results [5]. The flat panels tested had identical joint construction to the curved panel CVPB. For both panels, the maximum strain occurred at the inner skin surface. The ratio of the inner skin strain for the curved and flat panels was 1.14.

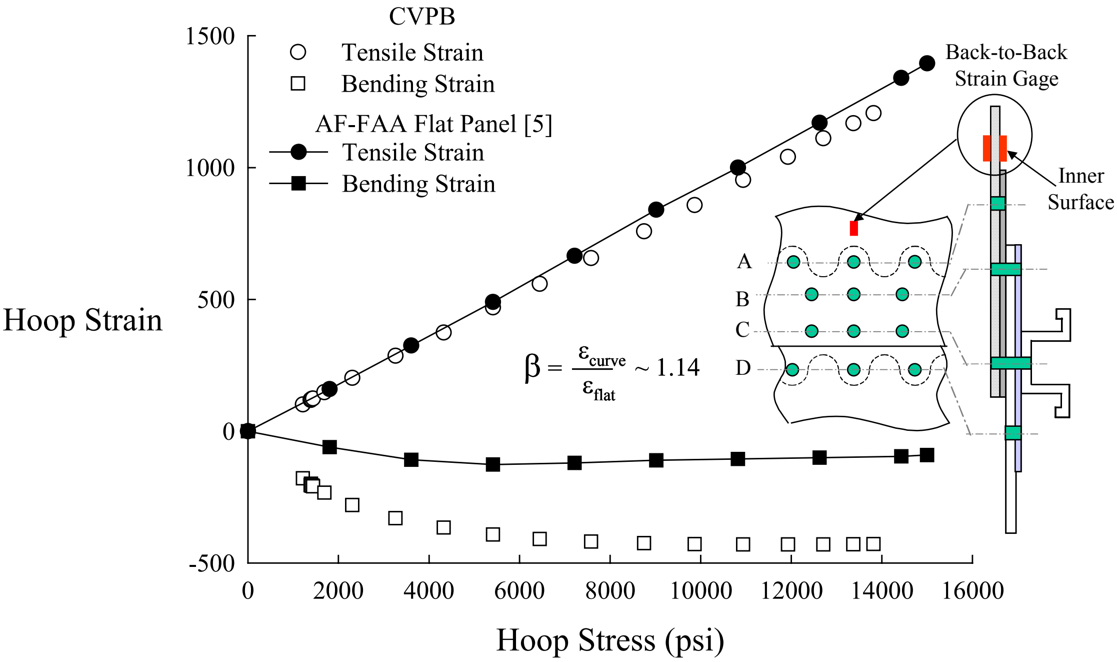


Figure 7. Strain near lap joint in curved panel CVPB and flat panel

The curved panel was then subjected to a fatigue loading. The typical damage evolution process in the outer row of the lap joint is illustrated in Figure 8 with a series of photographic images. The cycle number at which each image was taken is also shown in the figure. Damage was first observed in the rivet in the form of a rivet head crack. Subsequently, the crack grew along a curved path and seemed to follow the perimeter of the rivet stem, Figure 8(a). Water leakage from the crack indicated that it was a through-the-thickness crack. It should be noted that the loading used in this study was much higher than what a fuselage would experience during normal service conditions. The rivets are not designed to sustain such high fatigue loads. Thus, it is believed that the rivet head crack initiated at the rivet shank-countersink interface due to the stress concentration in that area and propagated upwards to the surface. As the fatigue test continued, a through-the-thickness crack appeared on the right side of the rivet at a distance from the edge of the rivet hole, Figure 8(b). The crack grew in both directions and eventually linked up with the rivet hole, Figure 8(c). At a later stage, another through-the-thickness crack appeared on the left side of the rivet, Figure 8(d). This crack also grew in both directions and linked up with the rivet hole, Figure 8(e). A similar damage evolution process was observed at the neighboring rivet, Figure 8(f). Eventually, linkup occurred, forming a large lead crack in the outer row, Figures 8(g) and (h).

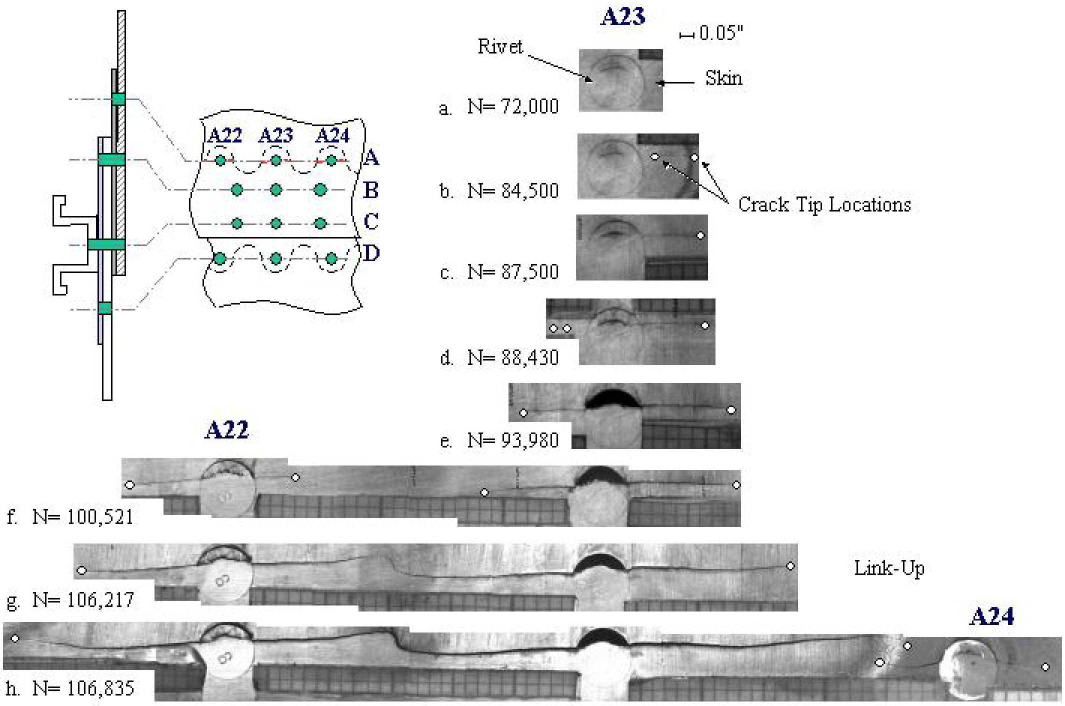


Figure 8. Crack growth process in outer critical rivet row

Crack growth rates measured for two sets of MSD cracks from rivets A22 and A23 prior to linkup displayed similar characteristics to other studies conducted by Fawaz [5] on flat panels and Piascik and Willard [7] on curved fuselage panels. Results obtained in this study (although for longer crack lengths) follow the trend of the data very well, as shown in Figure 9. This suggests that the fatigue cracks from these three different studies grew at similar rates.

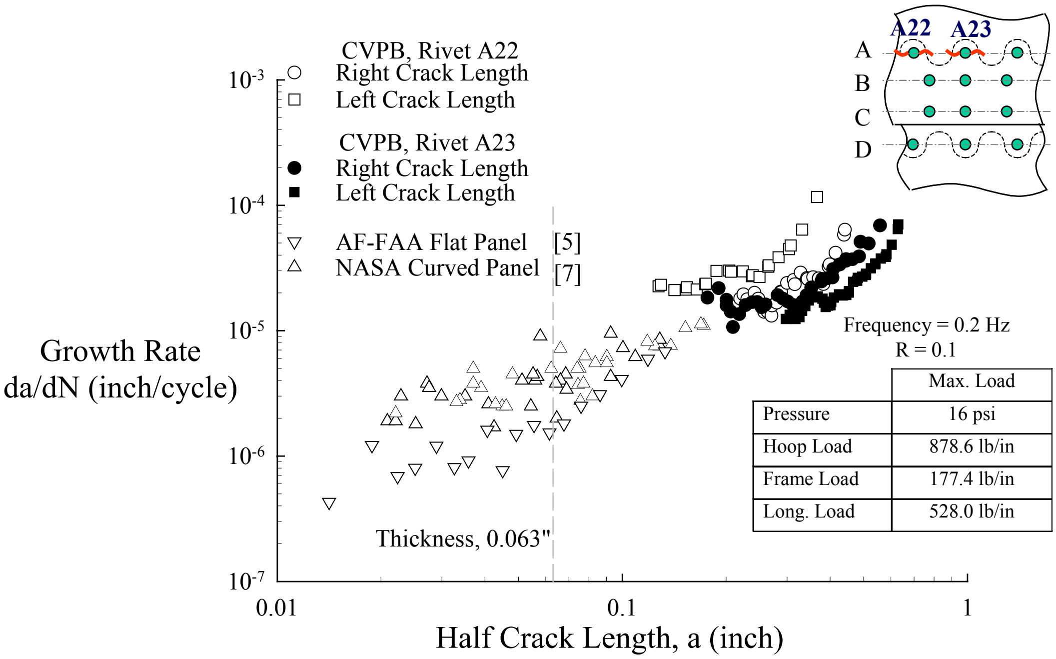


Figure 9. Fatigue crack growth characteristics prior to linkup

Using the engineering approach outlined in Figure 6, the analysis of the two sets of MSD cracks from rivets A22 and A23 prior to linkup was conducted. The flat panel crack growth rate data in Figure 9 was fitted to equation 1 to obtain the empirical SIF for the flat panel. A distribution of EIFS was then back calculated. Since the joints from the flat and curved panels were the same, it was assumed the crack initiation and small crack growth behavior would be the similar and that this EIFS distribution could be used for both panels (providing the same crack growth analysis and data are used). The strains shown in Figure 7 were used to scale the empirical curved panel SIF, equation 4. Using this expression along with the EIFS distribution, the curved panel crack growth analysis was conducted. Results are shown in Figure 10 in terms of the crack length as a function of cycles. The open symbols are the experimental data for the curved panel, and the solid symbols represent the flat panel data from Fawaz [5]. The solid lines represent analysis results for three EIFS: 0.0005, 0.001, and 0.0015 inch.

The MSD cracks eventually linked up to form a large lead crack, as shown in Figure 11. In the figure, a schematic is provided of the crack path along the outer rivet row between frames 2 and 3. The first crack linkup occurred between rivets designated A22 and A23 after 106,217 cycles. The lead crack then grew very

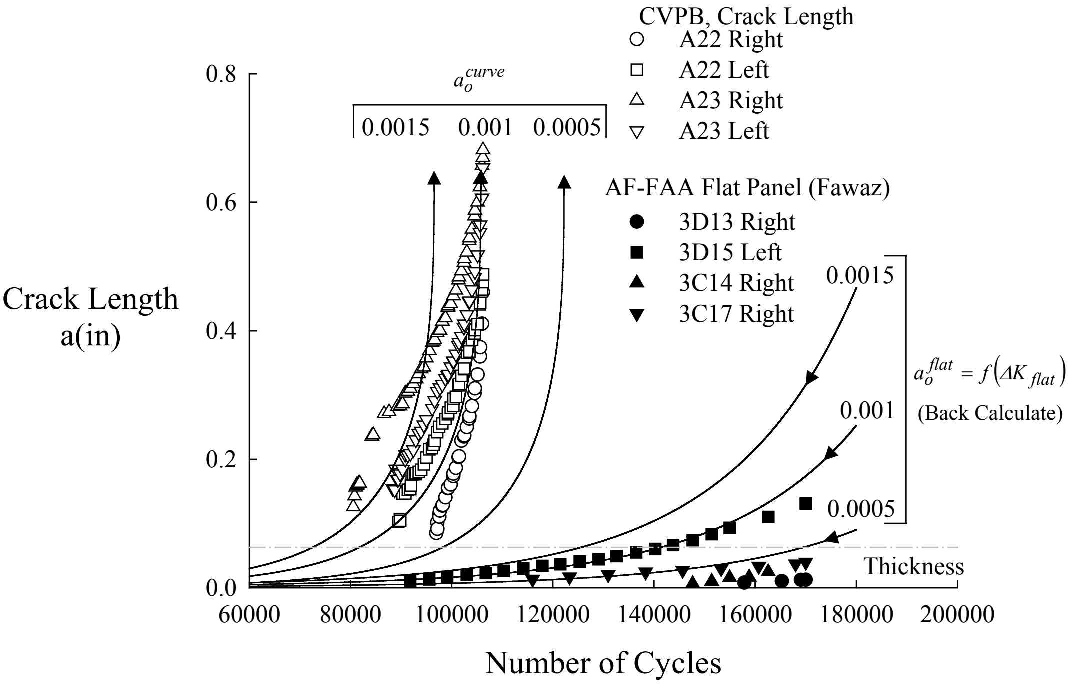
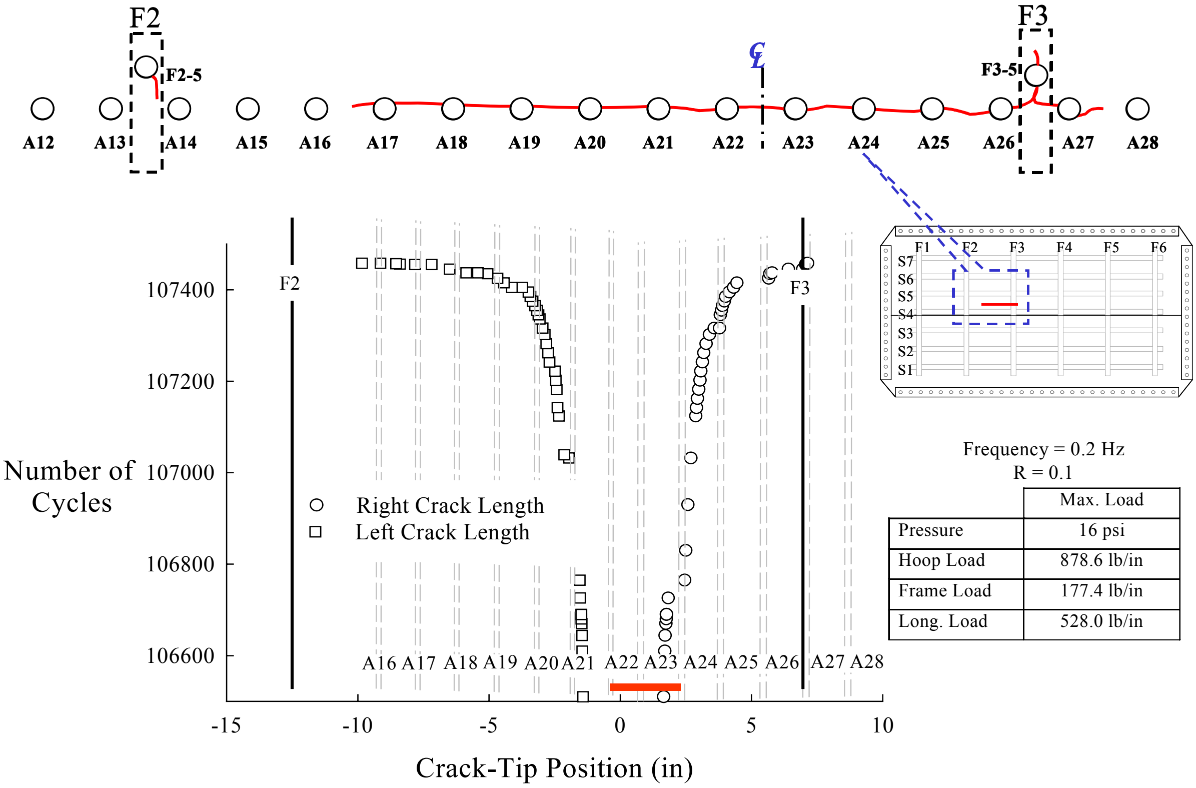


Figure 10. Fatigue crack growth analysis



Frequency = 0.2 Hz
R = 0.1

	Max. Load
Pressure	16 psi
Hoop Load	878.6 lb/in
Frame Load	177.4 lb/in
Long. Load	528.0 lb/in

Figure 11. Lead crack length as a function of fatigue cycles

rapidly. After 107,448 pressurization cycles, the MSD evolved to a 16" two-bay crack through rivets designated A17 and A27.

The panel was then subjected to quasi-static pressurization up to failure to measure the residual strength. The panel failed catastrophically at 17.8 psi pressure along the outer rivet row exhibiting no crack turning (flapping). As shown in Figure 12, the crack grew across five frames, designated F2 through F6. In addition, frames 3, 4, and 5 were fractured.

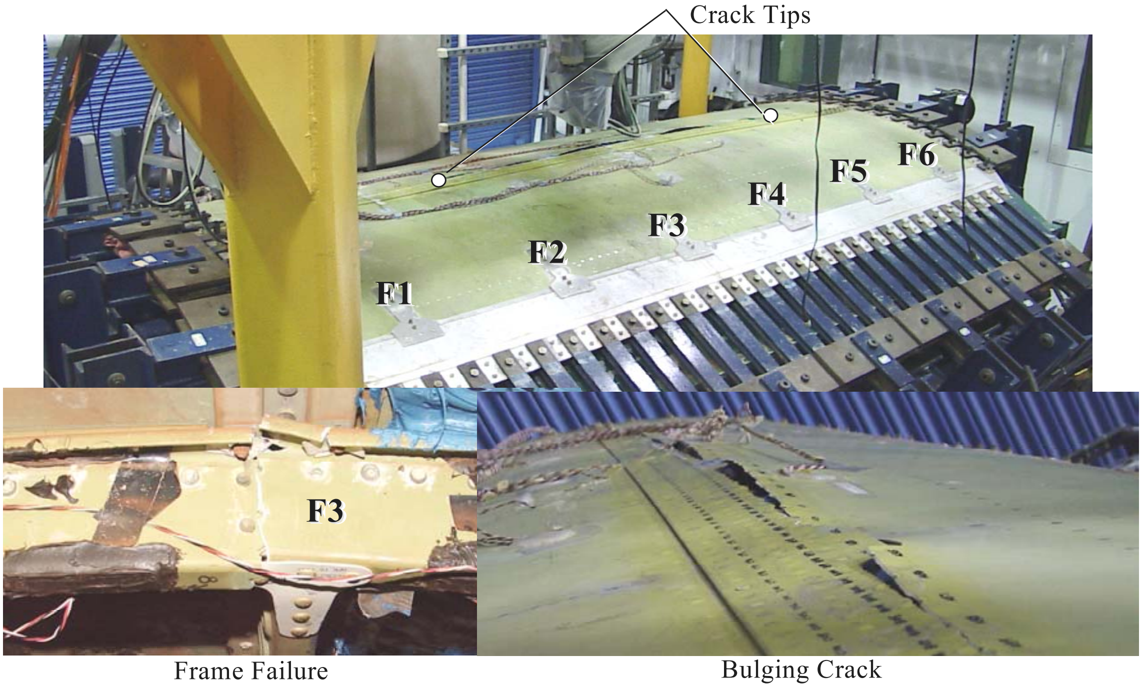


Figure 12. Failed panel with final crack

MSD Panel Test Results

The effects of multiple cracking on the fatigue crack growth and residual strength of large lead cracks were examined. Strains were first measured under quasi-static loading conditions to ensure proper load introduction to the panels. The strain measurements were highly repeatable and were in good agreement with the finite element analyses. The presence of multiple cracks did not affect the overall global strain response [2,4].

During constant-amplitude fatigue loading, symmetric, collinear crack propagation was observed. Reasonable agreement was obtained between experimental fatigue crack growth data and predictions relying on the effective SIF. Representative results are shown in Figure 13 for longitudinal lap joint panels CVP1 and CVP2 (contained multiple cracks). The data in this figure represent the lead crack tip position from the right and left sides as a function of the number of cycles. As shown in the figure, the presence of the MSD reduced the number of cycles to grow the lead crack to the final length by approximately 37%.

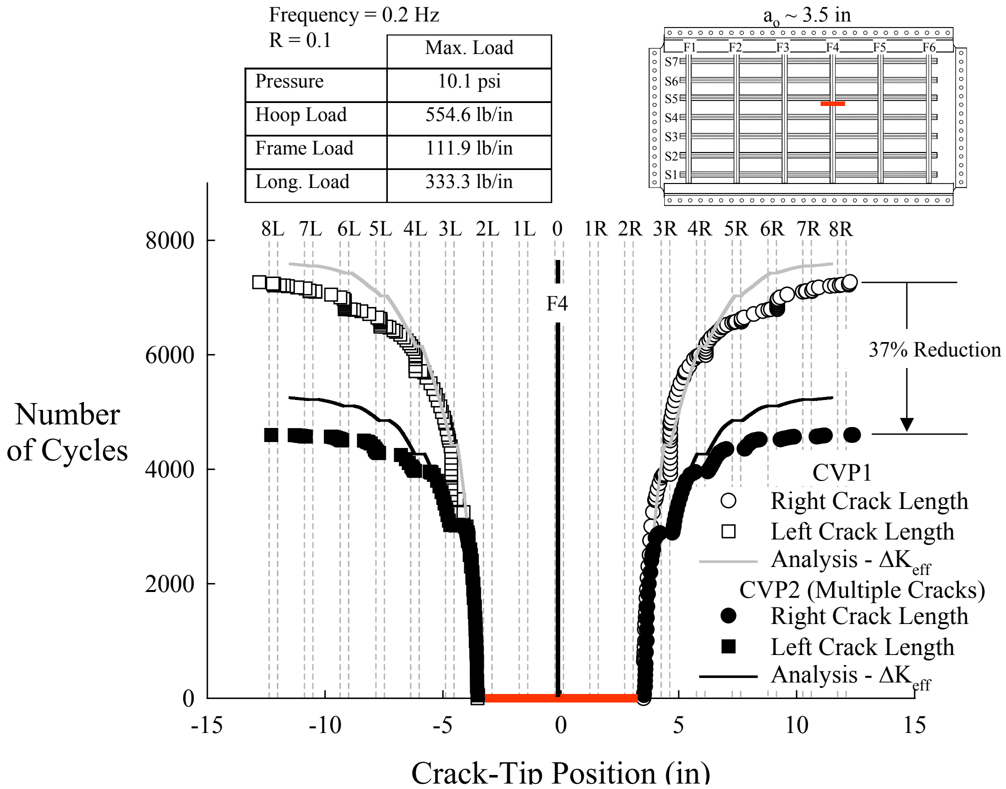


Figure 13. Fatigue crack growth data for panels CVP1 and CVP2

After fatigue testing, residual strength of the panels was measured. For the curved panels with the longitudinal lap splice, the initial damage consisted of a two-bay crack with a length of approximately 25″ with the central frame cut. Typical results from the residual strength test of panels CVP1 and CVP2 is shown in Figure 14. The data in this figure represent the crack extension for the left and right crack tips as a function of the applied pressure. During the test, cylinderical pressurization was applied quasi-statically, and the crack extension measured up to panel failure. As indictated in the figure, the presence of multiple cracks reduced the residual strength by approximately 20%.

Fatigue Enhancement Test Results

The ability of polyisocyanurate foam to reduce strains and enhance component fatigue performance was assessed. Several tests were conducted to determine the effect of this foam on the strain state, crack area out-of-plane deformation, and fatigue crack growth. Results showed that the strain reduced an average of 6.4%

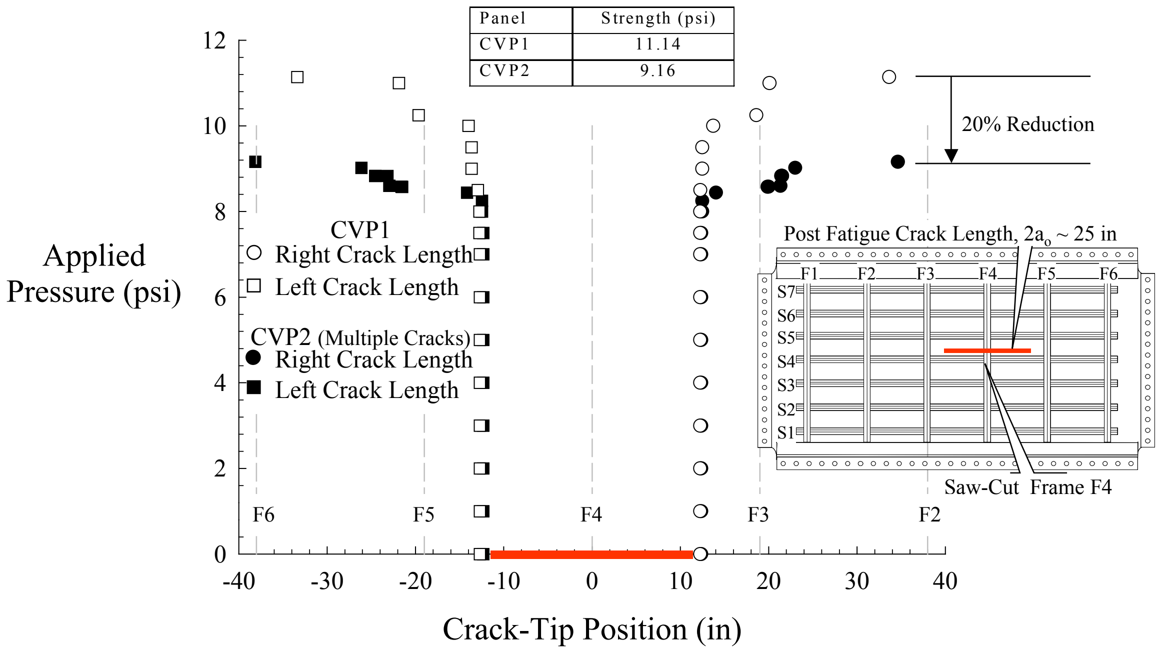


Figure 14. Residual strength data for panels CVP1 and CVP2

with the addition of the polyisocyanurate foam. In addition, the foam was effective in reducing the out-of-plane crack-bulging deflection, as shown in Figure 15. The data in the figure represent the out-of- plane displacements along the edge of the mid-bay crack. As expected, the out-of-plane displacement is highest at the centerline, and the shape of the crack-bulging profile is symmetric about the centerline. A 50% reduction in the out-of-plane displacements was measured at the crack centerline due to the polyisocyanurate foam.

The polyisocyanurate foam was effective in improving the fatigue crack growth behavior, as shown in Figure 16. The data in the figure represent the crack length as a function of pressurization cycles under constant-amplitude fatigue loading for the panel with and without foam. For both cases, the crack extension from the two crack tips is similar and collinear, indicating a uniform load in the region of the crack. The number of cycles to grow a fatigue crack by 0.65" was increased by 250% due to the presence of the polyisocyanurate foam. The solid line is the prediction made for the panel without foam using the SIF determined through finite element analysis. As shown, good agreement with experiments was obtained using the approach.

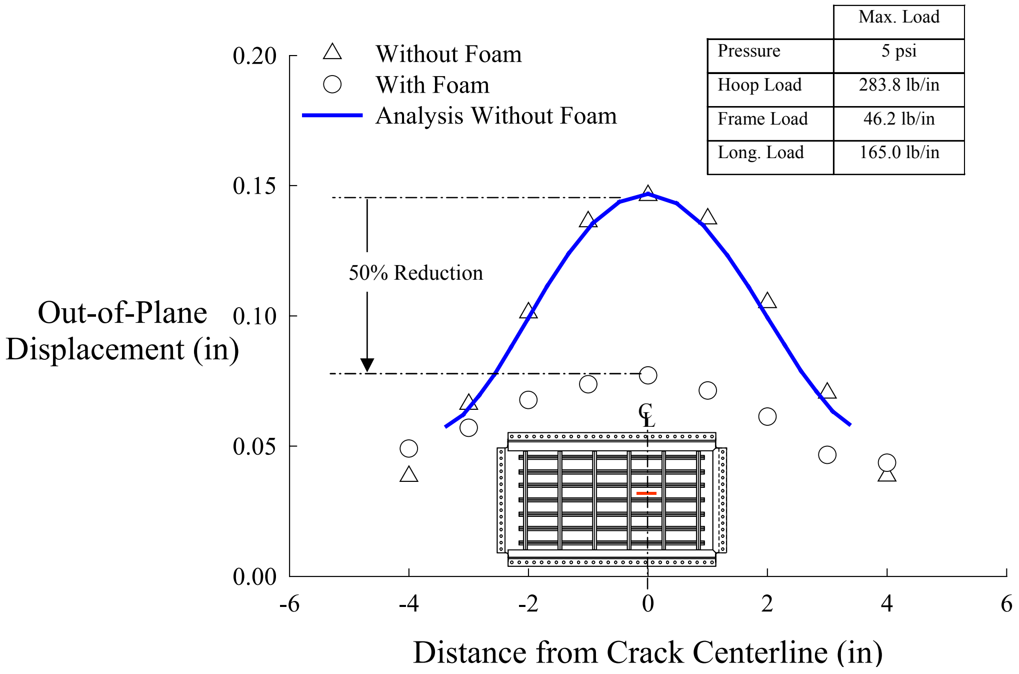


Figure 15. Crack-bulging profile for panel with and without foam

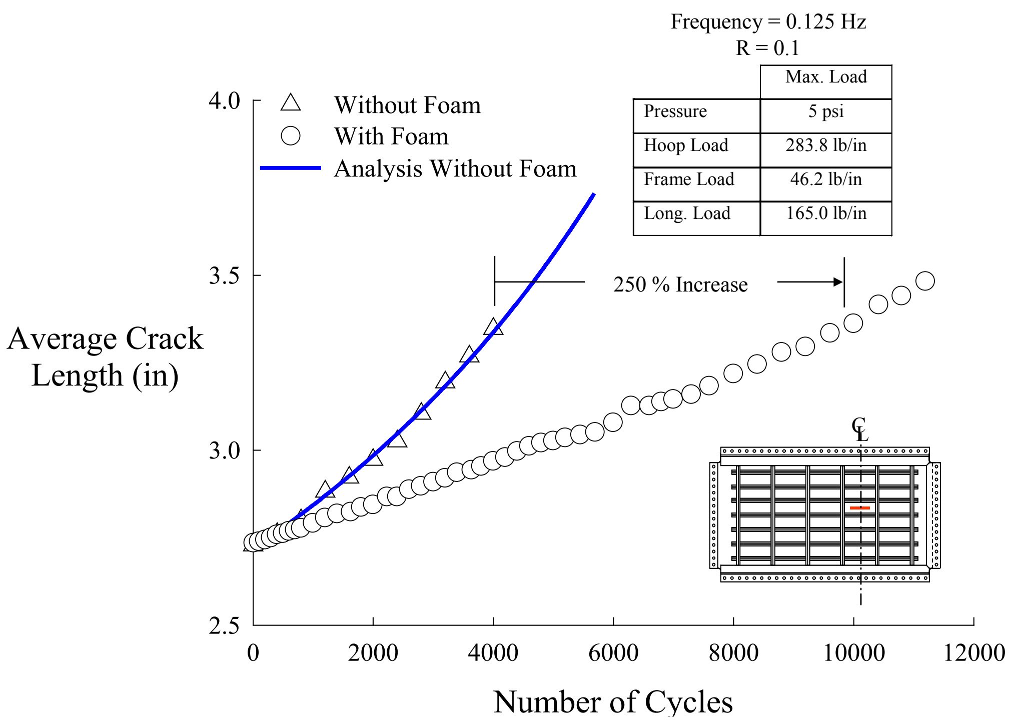


Figure 16. Fatigue crack growth for panel with and without foam

CONCLUDING REMARKS

The Federal Aviation Administration has characterized the fatigue crack growth behavior and residual strength of several aircraft fuselage structures. Both experimental and computational simulation methods have been used to study the evolution and development of multiple-site damage (MSD), the effects of MSD on the residual strength behavior, and the methods to reduce fatigue-related problems. The Full-Scale Aircraft Structural Test Evaluation and Research facility was used to obtain key data to calibrate and validate computational simulation approaches. Results show that the majority of fatigue life was spent in initiating and forming cracks from the inner-faying surface at rivet holes in the outermost fastener row in the lap joints and progressed through the thickness. Once first linkup occurs, crack growth was very rapid. Although small multiple cracks did not have an effect on the overall global strain response, it significantly reduced the fatigue life and residual strength. Polyisocyanurate polymer was effective in reducing out-of-plane crack bulging and fatigue crack.

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